Europa Mission Configuration Update to Accommodate Maturing Instrument Designs

Matthew Horner
Jet Propulsion Laboratory
4800 Oak Grove Dr.
Pasadena, CA 91109
California Institute of Technology
Matthew.D.Horner@jpl.nasa.gov

Alexander Eremenko
Jet Propulsion Laboratory
4800 Oak Grove Dr.
Pasadena, CA 91109
California Institute of Technology
Alexander.E.Eremenko@jpl.nasa.gov

Matthew Gentile
Jet Propulsion Laboratory
4800 Oak Grove Dr.
Pasadena, CA 91109
California Institute of Technology
Matthew.T.Gentile@jpl.nasa.gov

***Abstract*—A mission to Jupiter's moon, Europa has been of interest to NASA and JPL since the Galileo mission's magnetometer data predicted the presence of a subsurface ocean. The planned Europa mission would be equipped with a suite of instruments to perform both remote and in-situ sensing, the scope of which ranges from gravity science to characterizing the surface composition, with one of the objectives being to confirm the existence of the subsurface ocean. As the selected instruments mature, the challenge has been to select and refine a spacecraft configuration that is flexible enough to accommodate these changes without degrading the scientific capability of the spacecraft. An increase in instrument volume and power have prompted the growth of spacecraft engineering subsystems, which include the solar arrays and the avionics module. These changes have the potential to cause obstructions to the instrument fields of view, stray light keep-out-zones, and thermal radiative fields of view. Many of these criteria are addressed by the implementation of a dedicated instrument platform. Payloads that are unique to the Europa mission require new approaches to instrument accommodation, such as coupling the ice penetrating radar's radiating elements to the solar array. This paper will discuss these accommodation strategies for the Europa spacecraft to generate a new baseline design.**

TABLE OF CONTENTS

1. INTRODUCTION

Europa has been said to have the highest probability for finding life in our solar system beyond Earth. As such, it has garnered a great deal of interest within the science community over the last 20 years. In December of 2012, Hubble's ultraviolet instrument observed what scientist believed to be plumes of liquid water erupting from the moon's surface. Ever since, the possibility of a mission to Europa to interrogate the plumes and determine the composition of the suspected liquid water ocean has been a priority of NASA. In January of 2016, Hubble again turned its gaze to Europa to find yet again what is believed to be liquid water plumes extending hundreds of miles off of the surface. NASA's Jet Propulsion Laboratory, in a partnership with the Johns Hopkins University Applied Physics Laboratory has been developing a mission concept over the past several years, known as the Europa Multiple-Flyby Mission (formerly called Europa Clipper). Slated to launch on NASA's Space Launch System (SLS) in 2022, the project is nearing the completion of its formulation phase, and is looking to transition to the preliminary design phase in the beginning of 2017.

2. MISSION DESCRIPTION

Multiple Flyby Mission

The planned Europa mission would take advantage of a novel approach to explore Jupiter's smallest Galilean moon. To minimize the total ionizing dose (TID) seen by the spacecraft electronics, the mission utilizes multiple "flybys" of the moon, in which the spacecraft orbits Jupiter, and occasionally dips into the high radiation belt that encompasses Europa's orbit. The vehicle would perform a flyby of the moon over a multiple hour period where it takes the bulk of its critical science measurements. The flyby period begins around 66,000 km out from Europa, in the approach phase. The spacecraft points its imaging instruments to the center of the moon, called nadir pointing, and maintains pointing as the craft flies through closest approach (CA), coming as close as 25 km to the surface for many of the 40+ flybys. After CA, the spacecraft maintains nadir pointing as the departure phase takes the spacecraft out through 66,000 km from the moon. At this point the vehicle exits the flyby phase and continues on with its orbit of Jupiter, departing the radiation belt to spend multiple weeks downlinking date to Earth and pointing the solar arrays sunward to recharge the batteries in preparation for the next flyby.

During the flyby, it is critical that the remote sensing instruments housed on the spacecraft's nadir platform stay pointed at the surface of Europa. The y-axis of the spacecraft is the nominal pointing direction for all nadir instruments, and thus is pointed at the center of the moon in a maneuver called nadir pointing. As the spacecraft approaches the moon,

978-1-5090-1613-6/17/$31.00 ©2017 IEEE
This research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

reaction wheels provide a variable slew about the x-axis that increases in rate as the craft approaches the moon. When the spacecraft reaches CA, the z-axis is directly aligned with the velocity vector of the vehicle relative to Europa, known as the ram direction at CA (ram$_{CA}$). The instruments involved in in-situ measurements of the atmospheric dust and water plumes emanating from Europa (dubbed the ram instruments) are specifically aligned with the spacecraft z-axis to utilize this high velocity approach. The spacecraft coordinate system is defined in Figure 1.

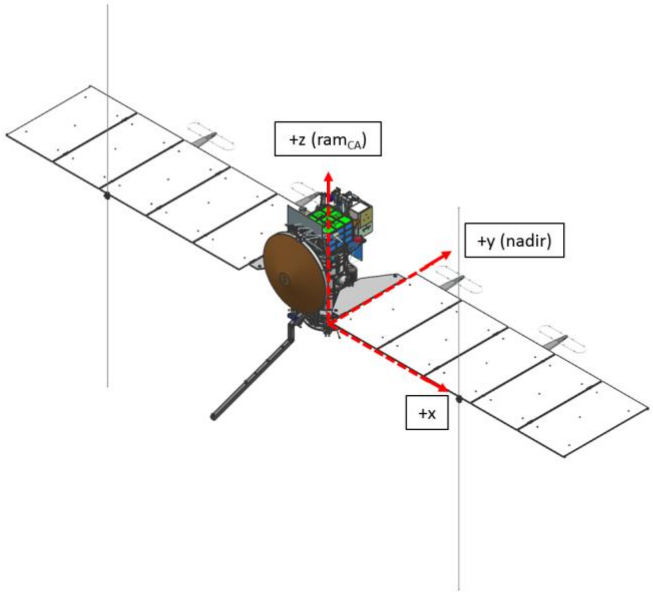

Figure 1 – Ram$_{CA}$ and nadir directions for configuration A5 in the nominal flyby configuration.

As the spacecraft exits CA and maintains nadir pointing, it continues to slew about the x-axis, at a decreasing rate. The sweep about the x-axis during the entire flyby means that the z-axis (ram$_{ca}$) moves through approximately 180-degrees relative to Europa. The result is that the ram instruments need to have a large effective field of view to take measurements for the majority of the flyby.

Direct vs VEEGA

The Europa mission relies on the development of NASA's next generation launch vehicle, the Space Launch System (SLS), to provide a direct trajectory to Jupiter. As shown in Figure 2, the thrust capability of the SLS Block-1 enables a 2022 direct trajectory to the desired Jovian orbit in approximately 2.5 years. By halving the transit time to the outer planets as compared to existing launch vehicles, the spacecraft's lifetime requirement can be significantly reduced. Other benefits over using alternative launch vehicles include returning scientific and engineering data earlier and reducing the spacecraft thermal design requirements.

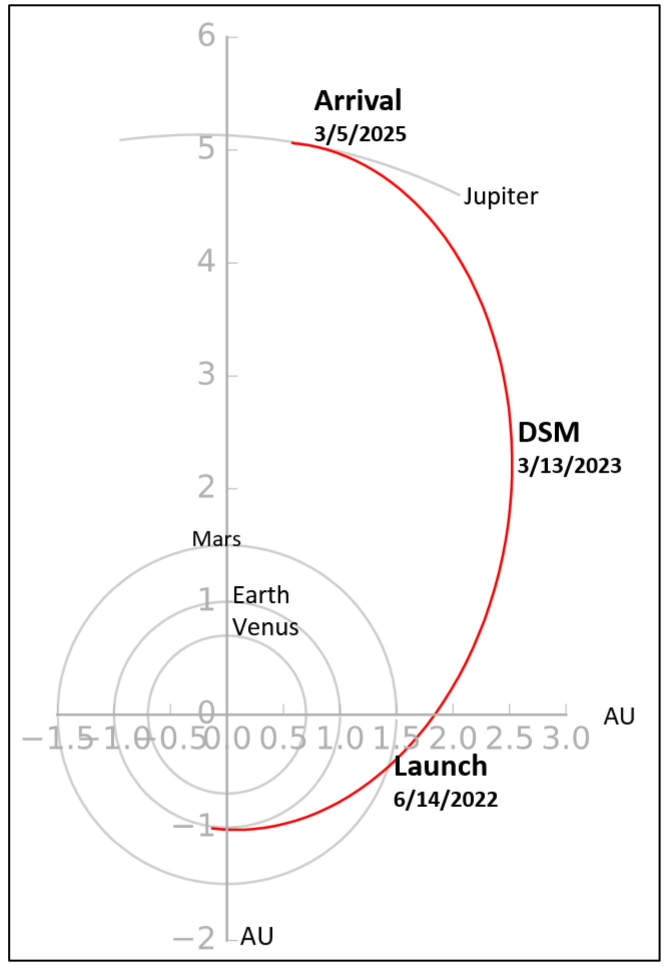

Figure 2 – Direct trajectory, 2022 SLS.

The backup launch vehicle for the Europa mission is the Delta IV Heavy, the most capable heavy-lift rocket currently in operation, but at a capacity less than the proposed SLS. As shown in Figure 3, utilizing the Delta IV Heavy would mean that a VEEGA (Venus-Earth-Earth Gravitational Assist) trajectory would be required.

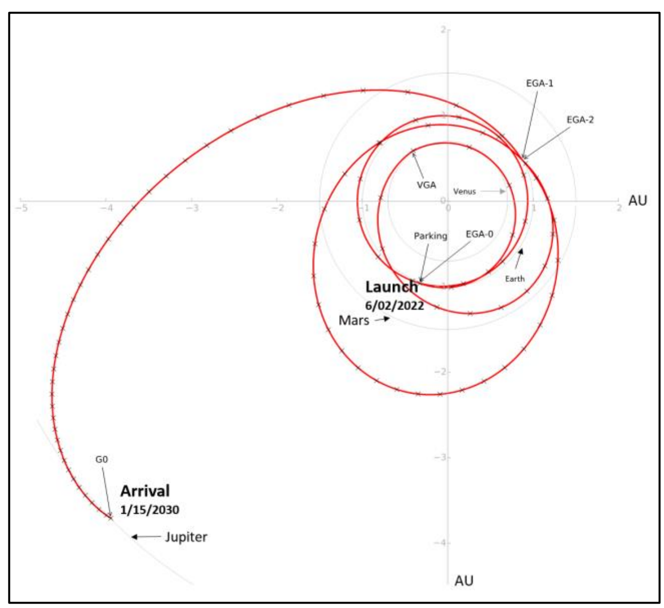

Figure 3 – VEEGA trajectory, 2022 Delta-IV Heavy.

These Venus and Earth flybys would be used to increase the velocity of the spacecraft on its way to Jupiter, but performing these gravity assists increases the transit time to approximately 7.5 years. Not only would this increase the total lifetime requirement placed on the spacecraft, but it would also involve a closer approach to the Sun, around 0.65 astronomical units (AU), as the vehicle flew by Venus, compared to the 1 AU minimum solar distance of the direct trajectory. This would represent a much higher demand on the thermal system of the spacecraft, and would involve additional measures to ensure the safety of the spacecraft's instruments and solar panels.

3. CONCEPT EVOLUTION

Selection of Configuration 2C

In December of 2015, the Europa project selected configuration 2C as the official baseline. This meant that the nadir and ram directions at CA were fixed with respect to the spacecraft coordinate system. Along with the determination of nadir and ram, configuration 2C had other major departures from the previous baseline. The high gain antenna (HGA) was reoriented and relocated from +z to –y. The REASON antenna was also moved from the top of the spacecraft to be symmetric about the deployed solar array. This was done for the purpose of satisfying the REASON antenna requirements, as symmetry of the spacecraft with respect to REASON, especially the large solar arrays, was desired in order to mitigate the effects on the antenna pattern. To achieve this symmetry, the solar arrays were relocated from the bottom of the vehicle, to reside nearer the center of gravity of the spacecraft. This had the additional benefit of balancing the spacecraft and lessening the effects of any solar array disturbances to the vehicle. Placing REASON behind the solar array (on the non-cell side of the panels) meant that the array could not be articulated to perform Sun searches. During each Europa flyby, the solar arrays are articulated to stay Sun facing, and provide power to the spacecraft at the time when the power demand is highest. The REASON instrument requires that the solar arrays are parked during the REASON science portion of the flyby (approximately 1,000 km on approach and departure from CA to Europa), but it was previously expected that the solar arrays would be articulated during all other portions of the flyby. During the rest of the orbit of Jupiter, the solar arrays would occasionally be articulated as the HGA remains pointed to the Earth, and various instruments perform calibrations. Fixing the solar arrays was a configurational constraint that was a major concern for the power generation story of the mission. After configuration 2C was approved as the project baseline in December of 2015, a tiger team was formed in January 2016 to address the accommodation of the REASON and MISE instruments.

2CH-SA

From January to March of 2016, the project selected tiger team evaluated several options for addressing the science concerns of REASON and MISE. Finding a suitable accommodation for the REASON instrument that minimized the spacecraft's effect on the antenna pattern, and simultaneously avoided intrusions into the keep-out-zones of the other instruments proved difficult. On top of that, accommodating a MISE instrument with a stringent thermal requirement of keeping the infrared detector at or below 80 K meant that even minor intrusions into the effective field of view of its thermal radiators could decrease the instrument's performance to unacceptable levels.

REASON—The REASON instrument's antenna pattern is highly sensitive to the spacecraft configuration, and selecting the wrong configuration could lead to a degradation in its science return. During the course of the tiger team activities, they discovered that the REASON HF and VHF antennas were not required to be accommodated in the same manner. Up to this point, the instrument's VHF and HF antennas were accommodated by way of two deployable booms, each containing radiating elements that ran parallel to each other. The VHF antennas consisted of four, folded dipole antennas, while the HF was a single, 16 m monopole. Separating the antennas expanded the option space to accommodate REASON. An attractive option that soon surfaced was what became known as the "H" configuration. As shown in Figure 4, this configuration decoupled the HF antenna from the VHF antenna by splitting it into two, separate, 16 m dipole antennas, and reorienting the dipole axes to be perpendicular to the direction of the VHF folded dipoles. This meant that the total length of the HF antenna had doubled. The deployed HF antennas, symmetrically placed on either side of the REASON boom resembled an "H" and the name was adopted to distinguish the major change.

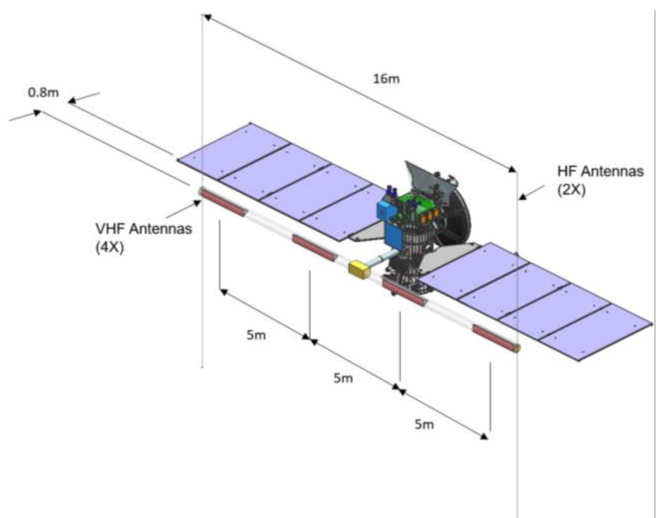

Figure 4 – Preliminary REASON Tiger Team "H" configuration mounted on a dedicated boom.

Multiple options for REASON accommodation were studied. Most consisted of deployable booms with VHF Yagi antennas spaced along the boom, and HF deployable antennas at the ends. Different configurations were explored in order to address the areas of instrument field of view and stray light keep-out-zone obstructions, solar array articulation envelope violations, and spacecraft controllability and dynamic

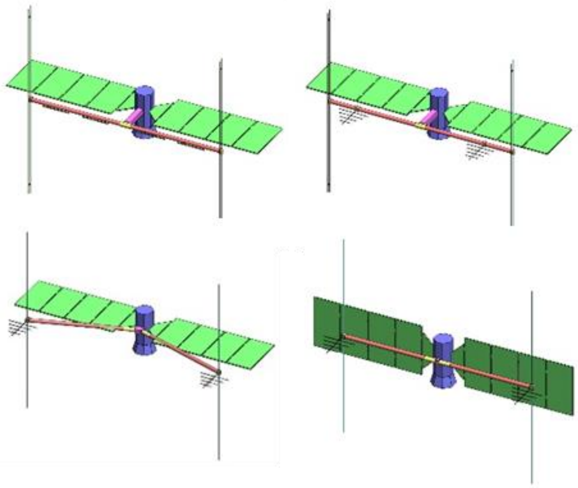

Figure 5 – Examples of alternate configurations explored for REASON accommodation

stability. Of utmost concern was the effect of the various configurations on the performance of the REASON antennas.

During the study, the size of a potential VHF Yagi antenna was approximated, and the result was an antenna that was on the order of 2 m wide by 4 m tall. Accommodating antennas of this size proved to be unachievable due to the spacing requirements of the antennas, and the amount of obstructions they created into other instruments stray light keep out zones.

MISE—The MISE instrument has faced challenges in achieving the 80 K infrared detector requirement, even before the instrument was selected. The notional payload assumed an instrument dubbed SWIRS (Short-Wave InfraRed Spectrometer) and even when the spacecraft's power subsystem was made up of Radioisotope Thermo-Electric Generators (RTGs), SWIRS anticipated issues with interactions with the spacecraft, and the other solar system bodies, most notably Jupiter and the Sun. Finding a suitable location for MISE that minimized it's thermal view to the spacecraft, and shaded the instrument from the Sun during its critical flyby activities remained a major challenge that was never solved during the tiger team's duration.

During the course of the team's study, multiple thermal architectures for MISE were explored. While passively cooling the instrument with a thermal radiator was the simplest implementation from an instrument complexity standpoint, other options offered better thermal performance. One such option was to employ a Winston cone architecture, in which a large cone acted as a Thermal shield, essentially rejecting any thermal inputs that were outside of the field of view of the cone (Figure 6). While offering the benefit of decreased thermal input (especially from the Sun), the Winston cone came with significant drawbacks when looked at from the overall configuration perspective. The size of the cone meant that accommodating it on the spacecraft would involve additional supporting structure, and the obstructions to the other instruments fields of view proved impossible to mitigate.

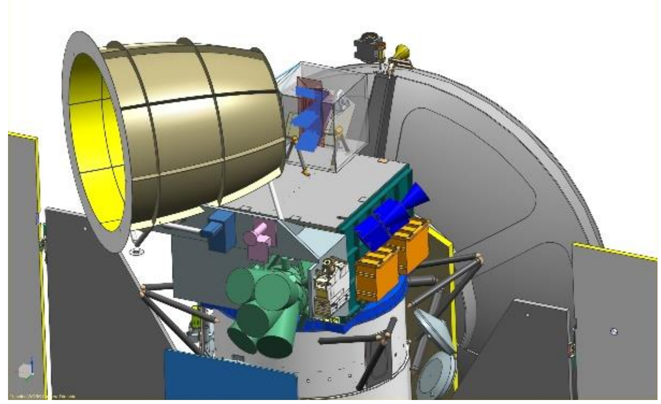

Figure 6 – MISE Winston cone architecture.

As the tiger team wrapped up activities in March of 2016, the MISE instrument team recommended the move to a dual cryo-cooler design as the baseline thermal architecture for the instrument. While presenting additional challenges to the spacecraft as a whole (most notably a higher power draw from the instrument, and a risk of microphonic inputs that could affect other instruments), the advantages of allowing a higher thermal radiator temperature, and being more robust to solar inputs to the instrument meant that cryo-coolers were an intriguing possibility, and were deemed as an architecture to explore for the next configuration.

Relocation of REASON to the Solar Array—At the conclusion of the tiger team activities, two options were presented (one primary, one backup) as suitable cases for establishing a baseline and working the issues through the remainder of Phase-A. As shown in Figure 7, the primary configuration consisted of dual deploying booms which emanated from the spacecraft in plane with the solar arrays. The booms were angled slightly away from the spacecraft to form a "V", with two VHF Yagi antennas and two HF deployable dipole antennas at the end of each boom.

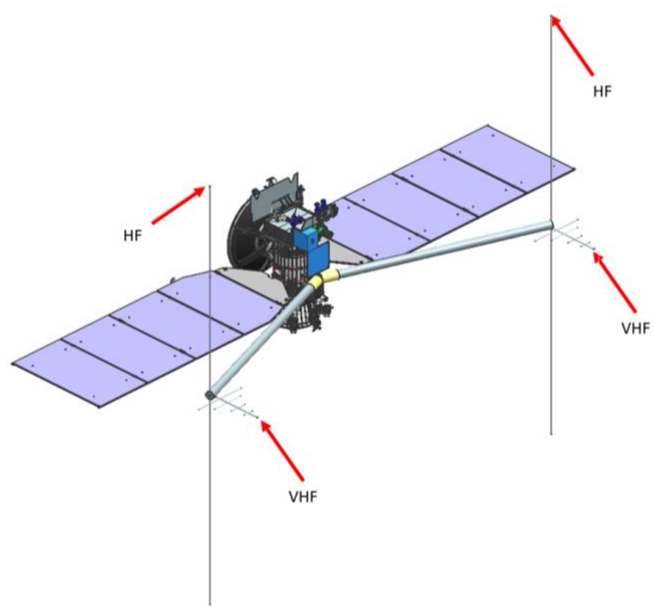

Figure 7 – Configuration 2CH-SA candidate, known as the REASON "V" configuration.

This configuration required modification to the inner solar array panels in order to allow for the full +/-180-degree articulation of the array. Despite the tiger team's recommendation, the project decided to adopt a configuration that mounted the REASON instrument (both the VHF and HF antennas) on the edge of the solar array as shown in Figure 8. This decision had a few major effects on the spacecraft as a whole. Most notably, it coupled the solar array and the REASON instrument development.

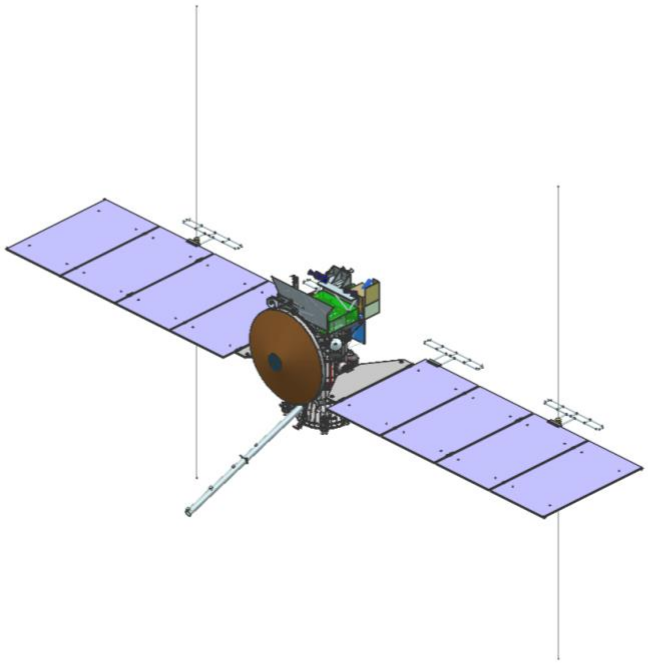

Figure 8 – Configuration 2CH-SA Deployed.

Configuration A4

Configuration A4, as shown in Figure 9, continued with the trend of mass and inertia growth of the spacecraft. The primary difference between configuration 2CH-SA and configuration A4 was the addition of a 5th solar panel per wing.

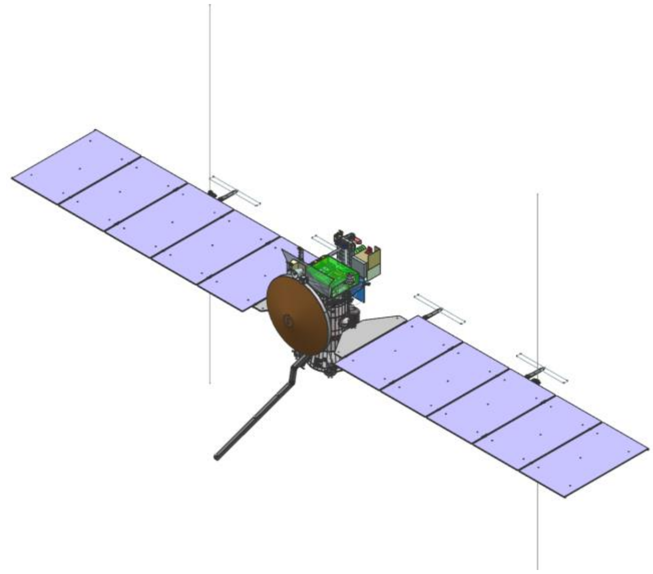

Figure 9 – Configuration A4 Deployed.

There were also reasons to rethink the accommodations strategies of certain instruments, namely MISE and ICEMAG. Changes to the instruments include removing MISE from the nadir platform, implementing a two hinge magnetometer boom concept to meet the tight ICEMAG knowledge requirements, and removing the 250 kg payload and associated 50 kg scar mass. The remaining remote sensing instruments (EIS NAC, EIS WAC, E-THEMIS, and Europa UVS) are mounted to a redesigned nadir platform that is kinematically mounted to the +Y vault panel. The platform also accommodates two Stellar Reference Units (SRUs), sometimes referred to as star camera, that are co-boresighted for operational redundancy.

Solar Array—The addition of a 5th solar array panel per wing (increasing the total number to 10 panels), increased the total solar array mechanical substrate area from 72 m^2 to 90 m^2. This was required to maintain a 30% end-to-end power margin given an increase in power demand from the payload and engineering subsystems. At 9 m^2 each, the 5th panels are identical to the other four panels and are arranged in-line for a total deployed wingspan of approximately 27 m. While other solar array concepts exist, this configuration has the least impact to the optical, stray light, and radiative fields of view (FOVs) of the instrument suite. The impact of this change to the inertias of the spacecraft and to the Guidance, Navigation, and Control (GNC) subsystem will be discussed in Section 4.

MISE Removal from Nadir Platform—The MISE instrument has experienced quite a bit of growth (both in power and mass) from the originally selected instrument to its current state. The growth of MISE from a model payload to its current design is shown in Figure 10.

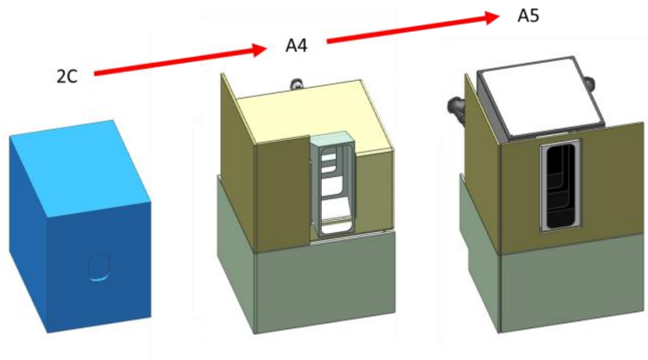

Figure 10 – Growth of MISE between configuration 2C and A4.

The operational parameters of an infrared spectrometer mean that the instrument thermal requirements are extremely difficult to meet. The MISE detector is required to operate at the lowest temperature of any instrument on the spacecraft, 80 K. This stringent requirement is challenging to achieve due to multiple factors, with the most notable being its thermal interactions with the spacecraft and the thermal input from the Sun during the flyby portion of the Europa science tour. During the approach phase of a flyby, it is critical that MISE performs a cool-down period in order to achieve the 80 K detector temperature required for spectroscopy. The Sun-Europa-spacecraft angle during many of the 40 plus Europa flybys is such that MISE receives a significant amount of solar-thermal input. To manage this thermal load, MISE updated their thermal architecture to include two cryo-coolers, each thermally strapped to dedicated L-shaped, thermal radiators. Two cryo-cooler/radiator combinations were required to cool the telescope and focal plane assembly. The L-shape form factor of the radiators (with radiating surfaces in the +y and +x directions) ensured that at least one surface of each radiator had an acceptable field of view to space. The presence of cryo-coolers meant that the effective radiating surface temperatures of the radiators was greatly increased from the 80 K detector temperature that was required by a fully passive radiator design.

The effect of the cryo-coolers on the mechanical design had negative implications on the configurational accommodation of MISE on the spacecraft nadir platform. Although they are designed to provide minimal vibrational disturbances (through mechanical motion cancellation and active vibration control), information available on the cryo-coolers indicated that a significant high frequency load could be produced and propagated through the instrument interface to the spacecraft. Seeing that the nadir platform was designed specifically to house the nadir facing instruments with high pointing accuracy and stability requirements, MISE posed a threat to these instruments whenever the coolers were operating. In addition, the radiator size and shape significantly increased the volume of the instrument, which meant that accommodation space for other instruments was decreased. The combination of all of these factors necessitated the decision to remove MISE from the nadir platform, and mount the instrument to the avionics vault, either directly, or via some to-be-designed structure to mitigate vibrational inputs to the spacecraft as shown in Figure 11.

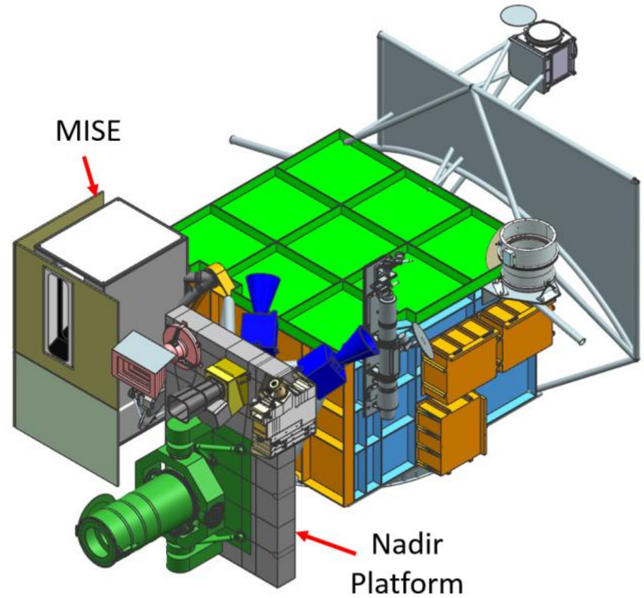

Figure 11 – Configuration A5 avionics module with MISE removed from the nadir platform and mounted directly to the +Y vault panel.

Magnetometer (ICEMAG) Boom Redesign—Prior to configuration A4, the 5 m magnetometer boom contained three hinges: two at the root of the boom for deployment and a third to separate the boom into two 2.5 m segments. To meet the 0.15 degree (3-sigma) knowledge requirement for the four ICEMAG magnetometer sensors, the hinge in the middle of the boom (elbow hinge) was removed. This increased the repeatability and stability of the boom at the expense of a more challenging volume required to stow the boom behind the HGA. With the boom being much taller in the stowed configuration, restraints on both the propulsion module and the avionics vault were added to the structure. A dog-leg (bend) in the boom was also incorporated into the design in order to clear one of four remote engine modules during deployment.

During the design update, a critical factor in the boom deployment was recognized: the initial deployment of the boom swept in front of the deployed solar array, and would be a mission critical failure should the boom deployment fail during its first phase. This realization meant that the deployment strategy for the boom would need to be sufficiently robust in order to provide the redundancy required of such a critical deployment. The details of the current magnetometer boom are discussed in Section 5.

NASA Probe Removed, Reaction Wheels Relocated—Prior to configuration A4, the Europa project was holding a burden of 250 kg of an unknown releasable payload (a NASA mandated probe) and 50 kg scar mass provided to accommodate it on the spacecraft. As the mass and complexity of the spacecraft design increased, NASA made the decision to remove the probe from consideration on the spacecraft. Prior to this removal, the spacecraft had not designed an interface to the probe. The strategy for accommodation was to leave free volume in a few key areas of the spacecraft, and to consider the 300 extra kg when computing the mass margin to the launch vehicle. The decision to remove the probe meant that accommodation volume was freed up for other hardware. On the top of the list for relocation were the spacecraft's four reaction wheels, which prior to configuration A4 were mounted to the +/-x sides of the propulsion module. The removal of the probe enabled the four reaction wheels to be mounted inside the skirt of the propulsion module on the bottom of the spacecraft. This was also an opportunity to ensure that the accommodation volume for the reaction wheels was increased to augment the configuration robustness to future wheel growth. Moving the wheels down to the bottom of the spacecraft also meant that the center of gravity (CG) height of the vehicle was reduced.

Vault Layout Update—The avionics vault concept for configuration A4 was based on the Mar Science Laboratory (MSL) chassis, where the electronic boxes are mounted on the base panel (-Z) of the vault as shown in Figure 12. While this concept reduces the complexity of System Integration and Test (SI&T) and thermal control, it requires the base panel to grow in response to new or revised electronics boxes. When assembling the vault, the lack of electronic boxes on the +Z panel allows it to be mated and de-mated to gain access without breaking the electrical connections of the contents. This concept also favors a "flat" rectangular form factor, which provides structurally inefficient mounting interfaces for the nadir platform, MISE, RF module and other secondary structures. Only utilizing one of six panels to internally mount components results in a 2D electronic box layout, which limits the benefit of boxes self-shielding one another from radiation.

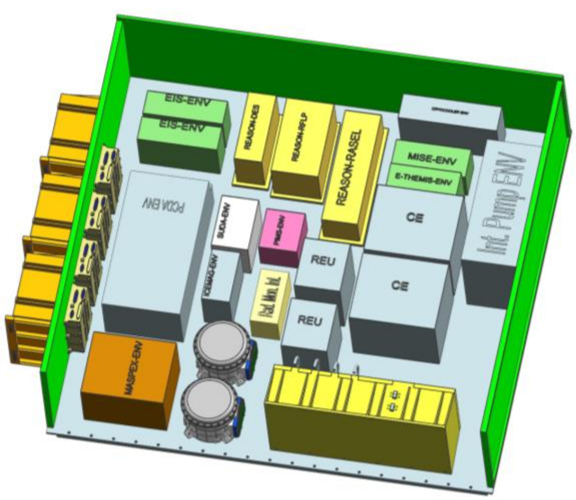

Figure 12 – Configuration A4 avionics vault concept based on the MSL rover chassis. External ribbing and RF panel removed for clarity.

These inflexibilities, along with the base panel expanding in the x-y plane to the point where it intersects with the restraints that support the stowed solar array, prompted a trade study to develop a new avionics vault concept for configuration A5. Other issues that were still present in the configuration A4 include: adjusting the location of the REASON HF and VHF antennas on the solar array to decrease obstructions of instrument stray light keep-out-zones, increasing the size of the reaction wheels to meet the torque and momentum demands of a larger spacecraft, and implementing a new avionics vault concept.

Configuration A5

Configuration A5, as shown in Figure 13, includes design updates to REASON, the avionics vault, and the reaction wheels to resolve open issues from configuration A4.

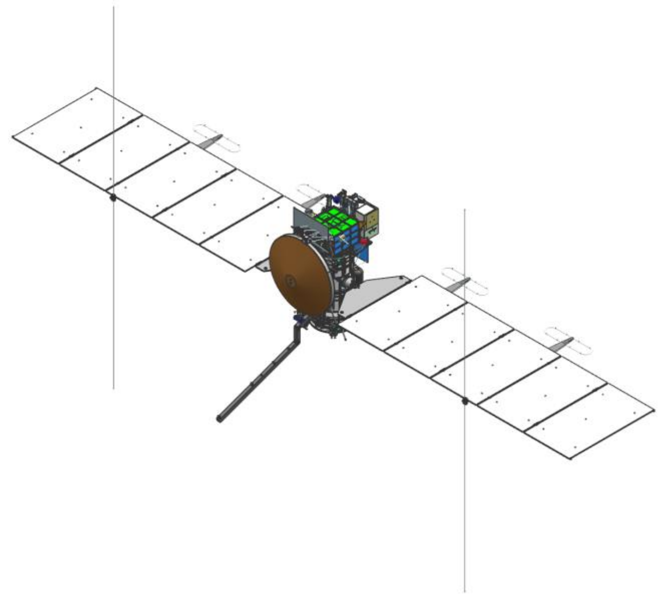

Figure 13 – Configuration A5 Deployed.

Major configurational changes to the REASON instrument and the avionics vault were the most notable. As a result of the changes, the spacecraft principal moments of inertia increased and the reaction wheel subsystem was re-evaluated for the first time since the project made the change from MMRTGs to solar arrays for its power system. Analysis showed that the reaction wheels were undersized and required additional resources to meet the performance requirements of the spacecraft.

REASON Design Update—Configuration A5 supports up to 74 kg of REASON hardware on the edge of the solar array, which causes the stowed frequency of each wing to drop to approximately 11 Hz. Analysis showed that the stowed frequency should be greater than 18 Hz to avoid large deflections that could damage the spacecraft during launch. The trade study and potential solutions are discussed in Section 4. The location of the four VHF antenna elements on the edge of the solar array also obstructs the stray light keep-out-zones of the EIS NAC, EIS WAC, and PIMs. The 5 m spacing between each VHF antenna element, known as the REASON 5m-5m-5m configuration (Figure 14), was implemented in configuration 2CH-SA to optimize the performance of the antenna array.

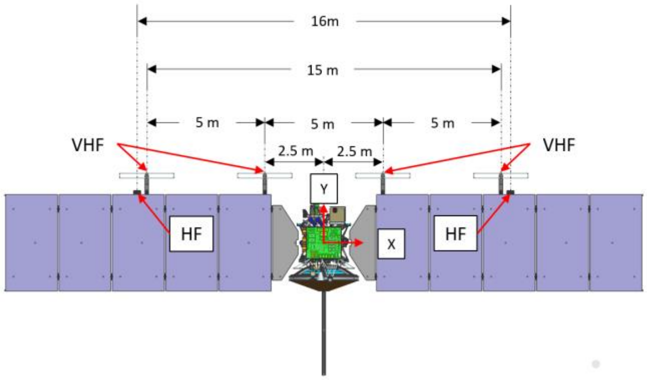

Figure 14 – Configuration A4 with REASON 5m-5m-5m VHF and 16 m VHF spacing.

By moving each VHF pair one meter outward on the solar array, the new 5m-7m-5m configuration (Figure 15) reduced the stray light keep-out-zone obstructions. The risk of glint (reflected light) from the VHF antennas damaging the nadir instruments can also be reduced applying a low specularity coating (e.g. black paint) to the antenna elements.

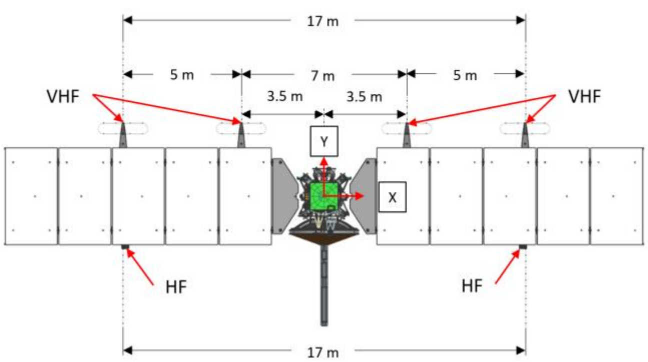

Figure 15 – Configuration A5 with REASON 5m-7m-5m VHF and 17 m VHF spacing.

Vault Concept Update—The avionics vault, which was previously based off the MSL chassis, was replaced with a "Dual Z" concept where the instrument and engineering electronics boxes are mounted on the +Z (top) and –Z (base) panels. This concept, as shown in Figure 16, results in a "cube" form factor, which improves the efficiency of the mounting structure of the nadir platform and minimizes the footprint in the x-y plane to avoid interfering with the stowed solar array restraints. Utilizing both panels also increases the packing factor of the vault compared to the previous design, which improves the self-shielding effect between boxes from radiation.

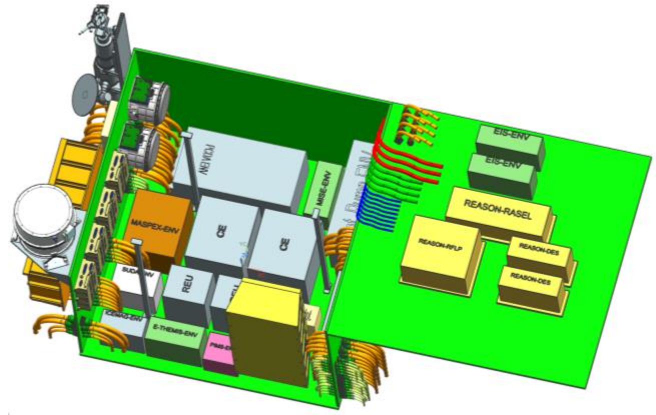

Figure 16 – Configuration A5 "Dual Z" vault concept. External ribbing and RF panel removed for clarity.

To improve accessibility of the vault for SI&T, the top panel would be opened via a "hinge" at the interface of the –X and +Z panels. The hinge will be ground support equipment that will be removed from the vault for testing and launch, thus simplifying the design and avoiding a mass impact to the spacecraft. Mounting boxes on the top panel complicates the thermal control of the vault, as it is difficult route the thermal fluid loop across the hinge line. Also, if the +Z panel must be removed during SI&T, the thermal loop will have to be disconnected and then tested after re-mating the panel. The schedule and cost associated with implementing a fluid loop on the +Z panel may be avoided by implementing passive thermal control, such as louvers, but this will reduce the heat retained by the thermal fluid loop.

Reaction Wheel Size Update—The reaction wheel design consists of four wheels arranged in a rectangular pyramid that are mounted inside the skirt of the propulsion module. In response to adding the 5th panel to the solar array, the momentum capacity for each of the four reaction wheels was increased from 50 Nms to 150 Nms each. Three of the four reaction wheels are required to achieve the required slew (rotation) rates for science and operations, with the fourth for redundancy. To increase the agility of the spacecraft, the project is working to reduce the inertias of the spacecraft. This can be achieved by decreasing the power required by the spacecraft to decrease the size of the solar array or by implementing solar array configurations that locate the solar panels closer to the CG of the spacecraft. The latter may obstruct the FOVs of the instrument suite and will be investigated in Phase-B.

4. MAJOR TECHNICAL CHALLENGES

Mass and Inertia Growth

To meet its science objectives, the spacecraft must have the agility to track Europa with its instrument suite during each flyby. Other operational maneuvers include pointing the spacecraft's HGA towards the Earth for communication and orienting the solar arrays towards the sun. The resources required to perform these maneuvers depends on the moments of inertia (MOI) of the spacecraft, which are a measure of resistance to rotation about a given axis. Over the past two years, the decision to transition from a radioisotope power system to relying on solar panels to power the spacecraft has increased the inertias of the deployed spacecraft by an order of magnitude. The solution space to controlling the spacecraft include: decreasing the mass of the spacecraft, investigating alternative solar array concepts, increasing the size of the GNC subsystem, and decreasing the power demand to shrink the solar arrays.

Change from MMRTG to Solar Array—Before entering Phase-A in the Fall of 2014, configuration A2 utilized five Multi-Mission Radioisotope Thermoelectric Generators (MMRTG) to power the spacecraft. This form of power generation has been common for spacecraft that have gone beyond 1 AU, as solar flux decreases with the square of the distance from the sun. At a mean distance of 5.2 AU, the solar flux at Europa is approximately 25 times less than at Earth. However, as solar array efficiency increases and production costs decrease, using solar panels for large deep-space missions, such as JUNO, is possible.

Compared to MMRTGs, the drawbacks of solar arrays include creating FOV obstructions, increasing the deployed principal MOI of the spacecraft, and increasing the thermal load to the spacecraft. To minimize the impact to the instrument FOVs, the solar panel concepts throughout Phase-A divide the required mechanical substrate area between two solar array wings that are comprised of three to five panels each. The panels are deployed in-line, as limiting the width of the array provides more unobstructed space to accommodate the instrument FOVs. However, in-line arrays place the collective panel mass far away from the center of mass (CM) of the spacecraft, thus increasing the deployed inertias of the vehicle. To generate the same power as five MMRTGs at Europa, the solar array must support 64 m^2 of mechanical substrate area. This transition increased the dry spacecraft inertia about the y-axis (I_{yy}) by 870%. As a 3-axis stabilized spacecraft, the impact to agility of the spacecraft was mitigated by increasing the mass, volume, and power of the reaction wheels. The large view factors of the solar array also increase the radiative thermal load to the spacecraft, prompting the instruments, such as MISE, to increase the size of their radiators.

Phase-A Mass Properties—The growth of the principal MOI of the Europa spacecraft throughout Phase-A is shown in Figure 17. The increase in MOI between configuration A2 to A3 can be attributed to switching from MMRTGs to solar power before entering Phase-A. During Phase-A, the predicted "model payload" was replaced with the selected payload, which increased the power required by the spacecraft [2]. To maintain a 30% end-to-end power margin for the mission, the required solar array mechanical substrate area between configuration A3 and 2C increased from 64 m^2 to 72 m^2. The solar panel configuration also changed from three 3 m by 4 m panels per wing to four 2.2 m by 4.1 m panels per wing that were offset from the spacecraft body by a dedicated yoke. The width of each panel decreased by 0.8 m in response to FOV growth of the nadir instruments. Despite adding REASON to the solar array in February of 2016 in configuration 2CH-SA, the overall inertias of the spacecraft decreased due to the removal of the 250 kg releasable payload and associated 50 kg scar mass. In May of 2016, a 5th panel was added to the spacecraft in response to another power increase from the payload and engineering subsystems, increasing the total mechanical substrate area from 72 m^2 to 90 m^2.

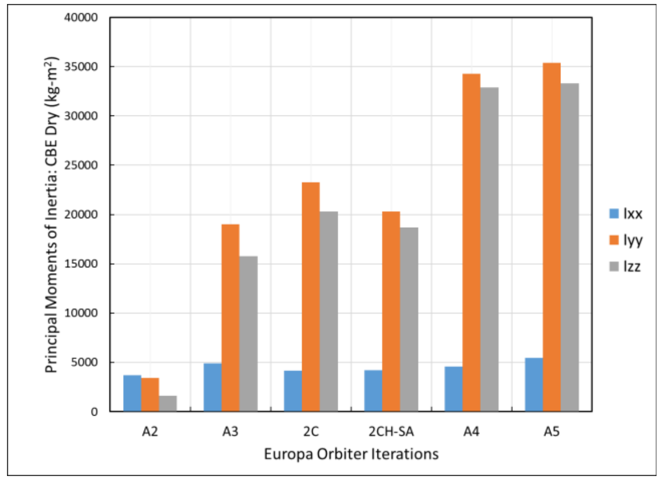

Figure 17 – Principal MOI for the dry current best estimate (CBE) spacecraft mass versus spacecraft configuration.

CG Location—Between configurations A3 and 2C, the solar array gimbal axis was translated in the +z direction on the propulsion module to decrease the distance between the CG of the solar array subsystem and the CG of the spacecraft. By the parallel axis theorem, this decreases the solar array's contribution to the MOI of the spacecraft. Placing the solar array closer to the CG also decreases the net torque on the spacecraft caused by input forces on the array, such as solar radiation pressure or excitation of their first deployed mode. Depending on the magnitude of the torque, this would cause the spacecraft to rock about its CG, which would require more resources for the GNC subsystem to control. This continues to be an issue as the mass of the avionics subsystem, which has a CG above the solar array gimbal axis, increases as the design matures.

The combination of CG above the separation plane and the wet mass of the vehicle also drives the required strength of the launch vehicle adapter. The baseline launch vehicle adapter is the ULA type D1666. With a CG height of approximately 2 m above the separation plane, the spacecraft is nearing the structural capability of the adapter. If the CG height or mass of the spacecraft continues to increase, a custom launch vehicle adapter will be required.

REASON Coupling to Solar Array

The decision to locate the REASON instrument on the solar array came with many of its own challenges. Marrying the power subsystem with arguably the most complex instrument on the spacecraft meant that the two systems would need to take part in a co-development program. One of the primary reasons for coupling the two systems was to fix the solar array with respect to the REASON instrument. During operation, the REASON antennas interact with the spacecraft, and the vehicle effectively acts as a part of the antenna. The other benefit of placing REASON on the solar array wings was to take advantage of the fact that the solar arrays could act as large booms that could place the REASON radiating elements far from the spacecraft body, to achieve the desired spacing requirements between the antennas. This meant that additional hardware was not required to deploy the antennas to their correct locations. However, the perceived benefits also came with a wealth of challenges that have continued to be worked, even up to the current design of the spacecraft.

REASON Stowed Frequency Issue—The inclusion of the REASON instrument on the spacecraft meant that additional mass would be added to the solar arrays. Not only would the antennas themselves need to be supported, but all of the cabling, heaters, matching networks, etc. would need to be accommodated to fully support the instrument. The addition of this extra mass (approximately 40 kg per solar array wing) meant that the dynamics of the spacecraft, in both the stowed and deployed configurations would need to be re-evaluated. In the stowed configuration, the REASON VHF and HF antennas were located on the +z and –z edges of the solar array, respectively. This additional mass at each end of the stowed solar array wings caused the frequency of the local area around the antenna to drop significantly. Local panel modes as low as 11 Hz were observed when the detailed finite element model of the spacecraft was evaluated. At that frequency, displacements of the VHF antennas on the order of 150 mm were seen, and clearance analysis indicated that hardware-to-hardware contact was possible. To address this concern, a study was performed to identify possible configurational changes to the solar array restraint scheme that could increase the local frequency of the panels and bring the antenna displacements down to an acceptable level.

Multiple options for increasing the local frequency were considered. The, seemingly, easiest option was to consider relocating the solar array restraint points in order to re-optimize the supports to the new solar array stowed configuration. This proved to have little effect on the local panel modes in the vicinities of the VHF and HF antennas, as the restraints were primarily designed to achieve the correct global stowed frequency of the solar array.

A promising option during the study was to consider additional restraint monopods that attached the spacecraft to the solar array in the vicinity of the HF and VHF antennas. This option provided the advantage of leaving the current solar array restraint geometry untouched, and directly addressing the problem areas on the solar array with dedicated hardware. Implementing this hardware meant that real estate and volumetric accommodations would need to be made on the spacecraft body as well as the solar arrays. For the VHF antenna restraints on the top (+z) of the vehicle, an attachment to the avionics vault provided the best load line to the stowed array. The attachment point on the array was placed between the two VHF antennas, and the monopod strut would be affixed to the solar array in the same manner as the primary solar array restraint hardware, by way of cup/cone hardware embedded into the panels, and a release mechanism attaching the array to the strut.

At the bottom of the spacecraft, the HF antennas were mounted on the bottom solar array edge. This meant that they were placed below the separation plane between the spacecraft and launch vehicle, and thus did not have any nearby structure to attach to. Attaching the solar array to the spacecraft body would mean that the load line for the strut would be inefficient, and would require additional structure to place the restraint in an ideal load bearing location. Implementing the additional REASON instrument restraints in the vicinity of the VHF and HF antennas proved to be a workable solution that addressed the local frequency and displacement issues of the antennas. The additional hardware increased the local panel modes to around 18 Hz, which brought the displacements down to acceptable levels, and ensured that loss of clearance, and hardware-to-hardware contact was not a possibility. Potential locations for the VHF and HF restraints are shown in Figure 18.

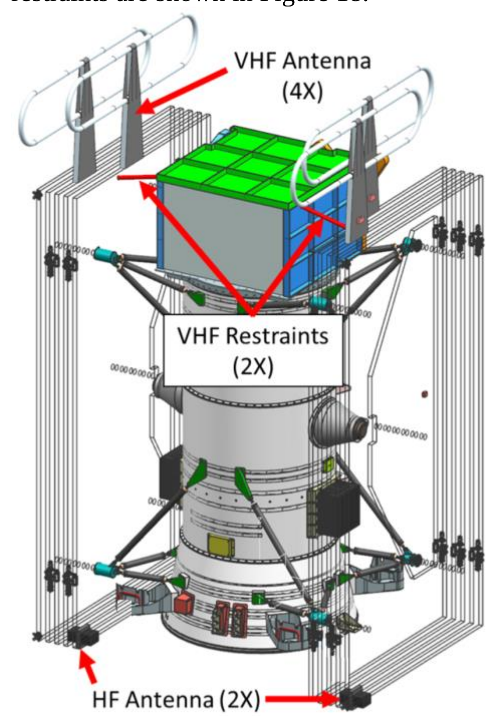

Figure 18 – Location of REASON VHF, HF, and VHF restraints for configuration A5.

The issue of adding additional hardware for the bottom (-z) HF restraints also led to the consideration of relocating the HF antennas to be co-located with the VHF antennas on the top (+z) side of the stowed solar array. An additional option with both antennas located at the top of the stowed array was evaluated, and it was determined that the VHF restraint hardware could provide the same benefits to the local frequency, even with the additional HF antenna mass located in the same vicinity. At the time of the writing of this paper, the Europa project is considering the move of the HF antennas to be co-located with the VHF, and it is likely that this configuration will be adopted for its mass savings by eliminating the HF restraint hardware and the complexity benefits of deleting a release mechanism per solar array wing.

5. DESIGN UPDATES

Nadir Platform Concept—The purpose of the nadir platform is to offset the remote sensing instruments (EIS NAC, EIS WAC, E-THEMIS, UVS) from the spacecraft to provide unobstructed optical, stray-light, and radiative FOVs. Two SRUs with diverse boresights are mounted to the back of the platform to meet the tight pointing and knowledge requirements of the instruments. The primary source of FOV obstructions include the 360° rotational envelope of the solar array and the REASON VHF and HF antennas. The configuration A5 nadir platform is kinematically mounted by three bipod pairs at an offset of 415 mm towards nadir from the +Y vault panel. Kinematically mounting the platform allows the structure to remain free of moments and distortions induced by temperature differences between the avionics vault, platform support structure, and instrument mounting platform. The instruments supported by the nadir platform is shown in Figure 19.

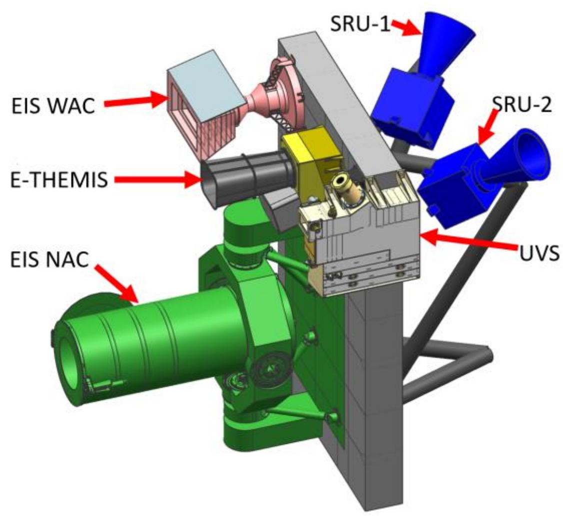

Figure 19 – Configuration A5 nadir platform instrument layout.

Mounting the nadir instruments on a common platform enables alignment of their baffle exit planes by placing each instrument on local brackets or by machining different mounting positions into the platform. This ensures that FOV obstructions between instruments are eliminated. Decoupling the platform from the spacecraft also allows design updates, such as increasing the size of the solar arrays or modifying the vault footprint, to be accommodated by increasing the offset of the nadir platform or by varying the length of the support structure.

Thermal Isolation—Prior to configuration A4, the nadir platform utilized active thermal control via a thermal fluid loop system. However, as the surface area of the platform grew to accommodate the instrument suite, it was determined that removing the platform from the loop would provide a net power savings at Europa. This is because the platform acts as a radiator for the interface between the avionics vault and the propulsion module, which must be kept above 20°C to prevent the propulsion system from freezing. Each instrument team also assumed that they would be thermally isolated from the platform, allowing them to tightly control their unit's temperature with a dedicated heater. This isolation scheme means that the fluid loop would not be able to harvest the heat from the instruments, resulting in it not being worth the additional cost, complexity, and power to keep the platform on the thermal loop. Assuming the platform will be constructed out of aluminum and covered by MLI to minimize heat loss, the platform is estimated to reach a steady state temperature of -60°C. If the backup VEEGA trajectory is selected, the nadir platform and nadir instruments will be protected from the high thermal flux at Venus by a sunshade.

Removal of the nadir platform from the thermal loop further complicates the structural implementation. To minimize heat loss between the avionics vault and the platform, the design goal for the total conductance across the support structure should be less than 0.2 W/K. This drives the material selection of the support structure to be made of GFRP, CFRP, titanium, boron, or other materials with a low thermal conductivity. Isolation may also be achieved by placing the platform farther away from the vault interface in exchange for an increase in mass and decrease in stiffness. Isolating the platform from the instruments and avionics vault creates large thermal gradients at the mounting interfaces. This will require additional structural and thermal analysis to avoid transferring thermal stresses to the instruments. This mounting scheme also makes it difficult to meet the knowledge and stability requirements of the instruments throughout the mission. Depending on the expected temperature variations, the use of a composite panel in place of a metallic structure may improve the dimensional stability of the structure.

Structural Implementation—The nadir platform is sized to support 130 kg, which includes the instruments, instrument harness, instrument mounting hardware, and MLI. With an approximate surface area of 1180 mm x 740 mm, the size and mass supported by the platform is similar to the Remote Sensing Platform (RSP) on Cassini. The RSP supports approximately 175 kg of hardware with 38 kg of structural mass. The ratio of structural mass to supported mass for the RSP was used to provide a preliminary mass estimate for the nadir platform. However, unlike Cassini, the nadir platform is offset farther from the spacecraft to avoid FOV obstructions created by the solar array. The nadir platform support structure is also structurally inefficient when compared to the RSP, as the mounting footprint available on the avionics vault is smaller than that of the RSP's. These two factors are why the mass estimate using the RSP ratio is treated as a lower bound for the nadir platform.

As shown in Figure 20, The nadir platform is comprised of three components: the instrument mounting platform, the support structure, and two tertiary brackets. The instrument mounting platform is assumed to be a 7075-T73 aluminum panel with integrally machined ribs. After machining, a closeout panel will be fastened to the panel to increase the specific stiffness of the design. When compared to a metallic platform, a honeycomb composite design may save weight and improve the dimensional stability of the platform when exposed to variations in temperature. The associated cost, schedule, and technical challenge of switching to a composite panel will be re-evaluated in Phase-B. The support structure is comprised of three CFRP bipods that are two inches in diameter with titanium end fittings. To approach a kinematic mount, all attachment points are assumed to use spherical bearings. The support structure material is subject to change, but these assumptions were used to estimate the mass and first fundamental frequency of the structure. The length of the bipods and the proposed materials also meet the thermal conductance goal of less than 0.2 W/K. The two tertiary brackets extend the platform to support the UVS and SRUs (not pictured) and will be refined in Phase-B.

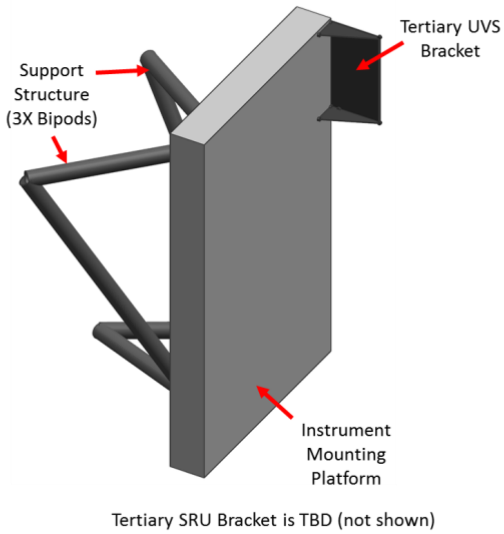

Figure 20 – Nadir platform concept.

The goal for the first fundamental frequency of the nadir platform and other secondary structures is 35 Hz. Sizing the structure to meet this goal prevents undesirable coupling with the launch vehicle modes and limits large fairing-to-payload deflections that could damage hardware. A simplified finite element model was created in NASTRAN to verify that the first mode of the nadir platform exceeded 35 Hz. The first mode of the structure is a torsional mode of the instrument mounting platform at 46.7 Hz, as shown in Figure 21. While this exceeds the 35 Hz goal, a 10 Hz margin is acceptable for Phase-A as the mounting interface and masses of the instruments are subject to change.

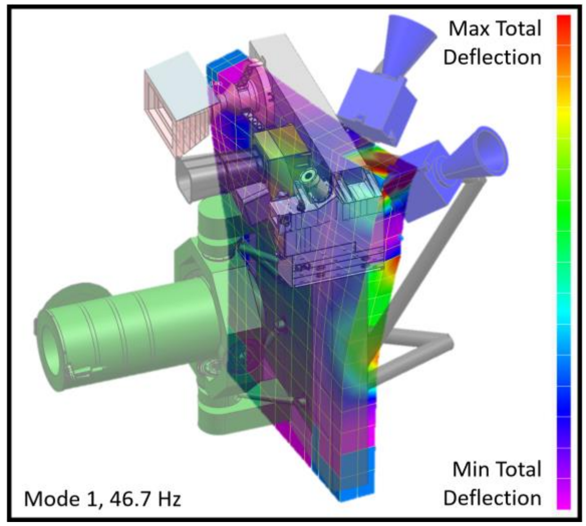

Figure 21 – Nadir platform preliminary frequency analysis.

Magnetometer Boom

The magnetometer boom has been updated in response to the tight attitude knowledge requirement of the ICEMAG instrument which resides on the boom. ICEMAG is comprised of four magnetometer sensors (two fluxgate, or FG, and two scalar-vector-helium, or SVH). The sensors are spaced along the boom, with the outer most sensor residing approximately 5 m from the body of the spacecraft (Figure 22). The purpose of this spacing is to allow the instrument to measure the magnetic field at multiple points, enabling the ability to determine the spacecraft's contribution to the magnetic field. This will allow ICEMAG to cancel out the spacecraft's field and isolate the fields of interest from Jupiter and Europa. Knowing the nature of the magnetic field around Europa will allow scientists to determine the physical makeup of the moon, and determine the salinity of the ocean beneath the icy surface. Because of the precise nature of the ICEMAG measurement, the exact orientation of each sensor with respect to the Jovian system needs to be known to a very high accuracy. One method for determining the precise attitude of the instrument is to measure the orientation of each sensor immediately after the deployment of the boom. By mounting an additional SRU to the boom, the location of the sensors can be determined despite misalignments during boom deployment. This measurement would be used in conjunction with the predictions of thermo-mechanical stability of the instrument throughout the science tour in order to calculate the total expected variations in the ICEMAG sensors orientations.

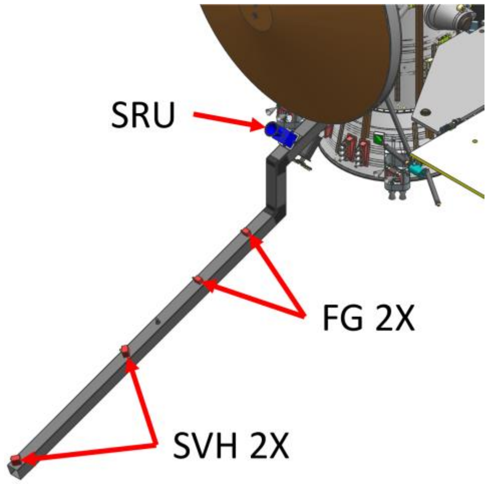

Figure 22 - Deployed magnetometer boom with ICEMAG sensors and SRU.

As an alternative, the mechanical subsystem has explored using precise deployment mechanisms with a high positional repeatability to eliminate the need for an additional SRU on the spacecraft. Departing from previous designs which had three hinges to successfully stow and deploy the boom, a two hinge design was employed. The deletion of the 3rd "elbow" hinge allowed for a decrease in the uncertainty of the deployment. This configurational change created an additional challenge in the stow and deploy scheme for the magnetometer boom. Without the elbow hinge, the boom would now have to stow behind the HGA with its full height extending approximately 2 m above the spacecraft avionics vault as shown in Figure 23.

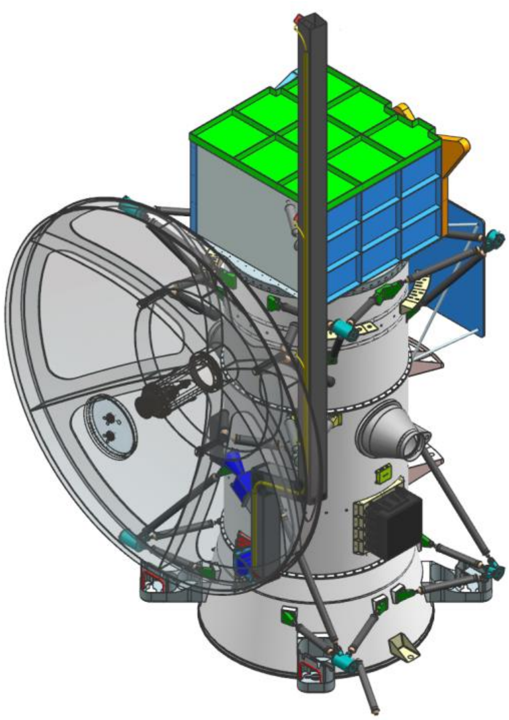

Figure 23 - Magnetometer boom stowed behind HGA (transparent). Solar array, instruments, and propulsion module structural details hidden for clarity.

Launch restraints were also required in order to restrain the motion of the boom during launch, and relieve loads on the deployment mechanisms. During the development of the deployment strategy for the ICEMAG boom, it was discovered that a mission-critical deployment was present. As the boom deployed from behind the HGA, its path of motion swept through the gimbal envelope of the solar array as shown in Figure 24. Not only would a deployment failure in the middle of this operation cause a permanent shadowing of the array, it would also limit the range of the solar array gimbal to effectively zero. The spacecraft would be constrained to the solar array orientation with the solar cells facing in the same direction as the HGA.

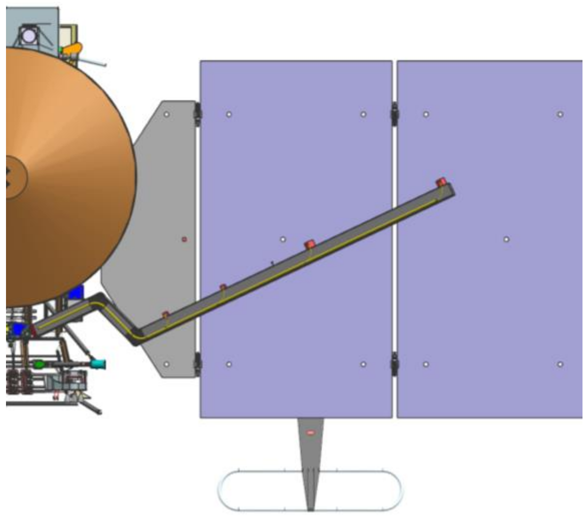

Figure 24 - Mission critical deployment portion of magnetometer boom deployment sequence.

This is the power generating configuration of the spacecraft, and it is different than the configuration needed for the REASON instrument to perform measurements during the flyby. REASON, being mounted to the solar array, would need to be pointed in the nadir (+y) direction as the spacecraft neared the closest approach phase of the flyby. On top of this, effectively fixing the solar array positon would degrade the power generating capabilities of the spacecraft as the arrays are normally gimbaled to track the Sun during nominal operation. The criticality of the ICEMAG boom deployment called for a robust deployment architecture. An active deployment, utilizing heritage dual-drive actuators was chosen for each deployment event. The dual-drive consists of two brushless DC motors driving redundant gearboxes. Each gearbox is driven by a single motor. The output shaft of the drive assembly can be driven by either motor/gearbox combination, either one at a time, or with both motors running simultaneously. This ensures that a single motor or gearbox failure will not preclude the operation of the drive assembly as a whole, as the other motor/gearbox can drive the boom deployment independently.

6. Conclusion

The development of the Europa spacecraft from the initial to current concept has had a number of challenges. Accommodating a suite of nine highly capable and sophisticated instruments, to operate in the extreme thermal and radiation environment at Jupiter is a task that presents a multitude of constraints and often conflicting requirements on the mission and spacecraft design. The current concept, configuration A5, represents a balance in achieving the mission's science and engineering requirements while attempting to keep the design and implementation of the spacecraft's subsystems as simple as possible. Although not free of issues and concerns that will continue to be worked, the spacecraft design is approaching a robust solution as the project's formulation phase draws closer to its conclusion, and looks ahead to the next stage in its preliminary design.

REFERENCES

[1] M. D. Horner and A. Eremenko, "Europa Spacecraft Configuration Optimization for the Solar Powered Vehicle," 2016 IEEE Aerospace Conference, 2016. doi: 10.1109/AERO.2016.7500810

[2] K. Lewis *et al*., "Use of Model Payload for Europa Mission Development," 2016 IEEE Aerospace Conference, 2016. doi: 10.1109/AERO.2016.7500708

BIOGRAPHY

Matthew Horner *received his bachelor's degree in Mechanical Engineering from the University of California, San Diego in 2006. He started at JPL in 2007 working as a design and integration engineer for the Mars Science Laboratory project. Over the past 9 years, he has developed hardware for various flight missions including SMAP and LDSD. He has been a member of the mechanical team for the Europa project for the last 3 years, and is currently the mechanical systems engineer for the project.*

Alexander Eremenko *received his master's degree in Aerospace Engineering from the Moscow Aviation Institute, Moscow, USSR (Russia) in 1984. He previously worked for Lavochkin Science & Production Association, Moscow, Russia for 11 years developing a variety of planetary and astrophysical missions. For the last 19 years Alexander has been working at NASA's Jet Propulsion Laboratory developing a variety of deep space missions including Ice&Fire, Solar Probe, Europa Orbiter, Pluto Kuiper Express, Mars Exploration Rover, Mars Science Laboratory, and SMAP. He is currently the Product Delivery Manager for the Mechanical subsystem of the Europa Multiple-Flyby mission.*

Matthew Gentile *received his B.S. in Mechanical & Aerospace Engineering in 2015 and his M.Eng in Aerospace Engineering in 2016 from Cornell University. He has previously worked as a Co-op at JPL on LDSD and the Europa Multi-Flyby mission and as a propulsion R&D intern at SpaceX for the Falcon 9 MVacD nozzle extension. He is currently the mass properties and a support engineer for the Europa Multiple-Flyby mission mechanical team.*